NASA Contractor Report 166099

Design and Fabrication of Realistic Adhesively Bonded Joints

Peter Shyprykevich
Grumman Aerospace Corporation
Bethpage, NY 11714

Contract NAS1-16952
May 1983

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

DESIGN AND FABRICATION OF REALISTIC ADHESIVELY BONDED JOINTS

by Peter Shyprykevich
Grumman Aerospace Corporation

SUMMARY

Eighteen bonded joint test specimens representing three different designs of a composite wing chordwise bonded splice were designed and fabricated using current aircraft industry practices.

Three types of joints (full wing laminate penetration, two-side stepped; mid-thickness penetration, one-side stepped; and partial penetration, scarfed) were analyzed using state-of-the-art elastic joint analysis modified for plastic behavior of the adhesive. The static tensile fail load at room temperature was predicted to be:

- 1026 kN/m (5860 lb/in.) for the two-side stepped joint
- 925 kN/m (5287 lb/in.) for the one-side stepped joint
- 1330 kN/m (7600 lb/in.) for the scarfed joint.

All joints were designed to fail in the adhesive.

INTRODUCTION

Recent studies (reference 1) have developed preliminary designs for advanced composite wing structure using bonded composite/metal construction. The objective was the development of a cost-effective structure with reduced weight and increased structural integrity. The design guidelines were:

- Maximum use of composites in the wing skins
- Emphasis on use of advanced metallic configurations for the substructure
- Elimination of mechanical fasteners penetrating the lower wing skin
- Use of concepts which exhibit potential low fabrication and assembly costs.

To carry the results of this study one step further, test specimens representing three different composite wing cover to metal spar-bonded joint concepts

were designed and fabricated. The specimens will be tested in static tension and fatigue at a later date to provide test evaluation of the joint capability in the wing chordwise direction.

The specimen bonded joints, both stepped and scarfed, display state-of-the-art realism in the selected material and configuration. Furthermore, the specimens were fabricated using current production techniques in processing composite material, bonding, titanium machining and welding, and quality control.

Many individuals at Grumman contributed to the work reported herein. The author wishes to acknowledge the efforts of Mr. Gordon Hudson in Design, Mr. Hans Borstell in Materials and Processes, and Mrs. Louise T. Coleman in Manufacturing.

SPECIMEN DESIGN AND ANALYSIS

Materials

The selected composite material was AS-1/3501-5A graphite/epoxy in 76.2-mm (3-in.) wide form with fiber volume in the cured state of between 58 and 62 percent. This material has been fully characterized at Grumman, design allowables are available, and it is being used on the Shuttle wing spars and X-29A wing covers.

Titanium (Ti-6Al-4V) was the choice for the metal adherend because of its low thermal coefficient of expansion and extensive use in aircraft structures. The use of aluminum was not considered because of its high coefficient of thermal expansion compared to graphite/epoxy and the potential for galvanic corrosion between noble and non-noble materials.

There are a number of adhesive systems in use that are applicable to bonded joints, their choice determined by temperature and environmental applications. FM-300K was selected on the basis of its shear compliance and its current use in the production for the F-18 wing.

Design

The adhesive composite-to-metal joints were designed to represent a realistic wing-type chordwise splice between a graphite/epoxy cover and metal spar. Realistic tension design limit loads would range between 438 to 700 kN/m (2500 to 4000 lb/in.), which translates into the ultimate design range of 657 to 1050 kN/m (3750 to 6000 lb/in.). Since the "B" basis allowable is usually taken as 0.8 on average for composites, the specimens were designed to fail between 823 and 1313 kN/m (4700 and 7500 lb/in.) at room temperature.

A major requirement of cover-to-substructure attachments is their ability to resist out-of-plane normal loads such as would result from fuel pressure in a wing box. The joint designs that were considered have spar caps embedded in the composite covers to improve the joint's out-of-plane or normal load capability.

Because the embedded spar cap reduces the chordwise continuity of the wing cover plies, the joint design becomes a compromise between normal load and chordwise load requirements. The joint designs, therefore, include spar caps embedded in two different levels: to mid-thickness and to the full thickness of the cover laminate. The latter represents a chordwise splice through the entire cover thickness and, as such, makes the wing cover fail-safe.

The cover thickness and layup were sized to be able to carry 2277 kN/m (13,000 lb/in.) in the wing spanwise direction. A typical wing laminate in the spanwise direction would consist of 40% 0°, 10% 90°, 50% ±45° direction plies. For this laminate, the allowable design ultimate stress for graphite/epoxy without notches is 586 MPa (85 ksi), which translates into a required thickness of 3.9 mm (0.153 in.). Therefore, for a representative chordwise wing splice, a 32-ply (assuming ply thickness of 0.133 mm (.00525 in.)) laminate consisting of 4/12/16 (4 plies in 0°, 12 plies in 90°, and 16 plies in ±45° directions) will satisfy all requirements. The strength of this laminate away from the joint is calculated to be 1820 kN/m (10,400 lb/in.) in the wing chordwise direction.

Plies of graphite/epoxy were added in the grip area of the specimen, in addition to the fiberglass tabs, to achieve the same load capability as the unnotched basic laminate and to preclude failure outside of the test section.

The specimen configurations that were obtained after several analysis/design iterations are shown in figures 1 through 4. The overall specimen geometry, figure 1, was dictated by aspect ratio and test machine requirements and is the same for all three concepts. The bonded-on-fiberglass tabs were designed to transfer the tension load by shear into the grippers. The required thickness of the fiberglass tabs were determined by assuming a 6.35-mm ($\frac{1}{4}$-in.) thick steel gripper plate.

The details of the three bonded joint concepts are shown in figures 2, 3, and 4. Joint Concept A, figure 2, is a proven, F-14 horizontal-stabilizer-type symmetrical-stepped graphite/epoxy-to-titanium bonded joint. This joint provides high pull-off strength and its manufacturing feasibility is well established. Furthermore, because the joint is symmetrical with respect to the laminate centerline, the analysis of the joint is straightforward and this concept is therefore a good baseline for comparison between test and analysis of the two alternate concepts.

Joint Concept B, figure 3, provides for partial chordwise composite skin continuity. The stepped design is similar to that of Joint Concept A but the partially uninterrupted skin makes it potentially a more practical and cost-effective joint, requiring less titanium and less machining or chem milling. The pull-off strength may be lower than that of Concept A, but its partial chordwise cover continuity and the reduced quantity of titanium used make it a more practical spar-to-cover joint for a multi-spar wing structure.

Joint Concept C, figure 4, has an embedded, scarfed, titanium spar cap. The well-established high strength and manufacturing feasibility of scarfed joints makes this concept a viable alternative. To ensure failure in the adhesive, a fairly large but still realistic scarf angle was chosen.

To provide the required spar web height, for all three concepts, the titanium caps (blades) were machined from 12.7 mm (0.5 in.) titanium stock and then electron-beam (EB) welded to a titanium plate approximately 254 x 140 mm (10 x 5.5 in.) in size.

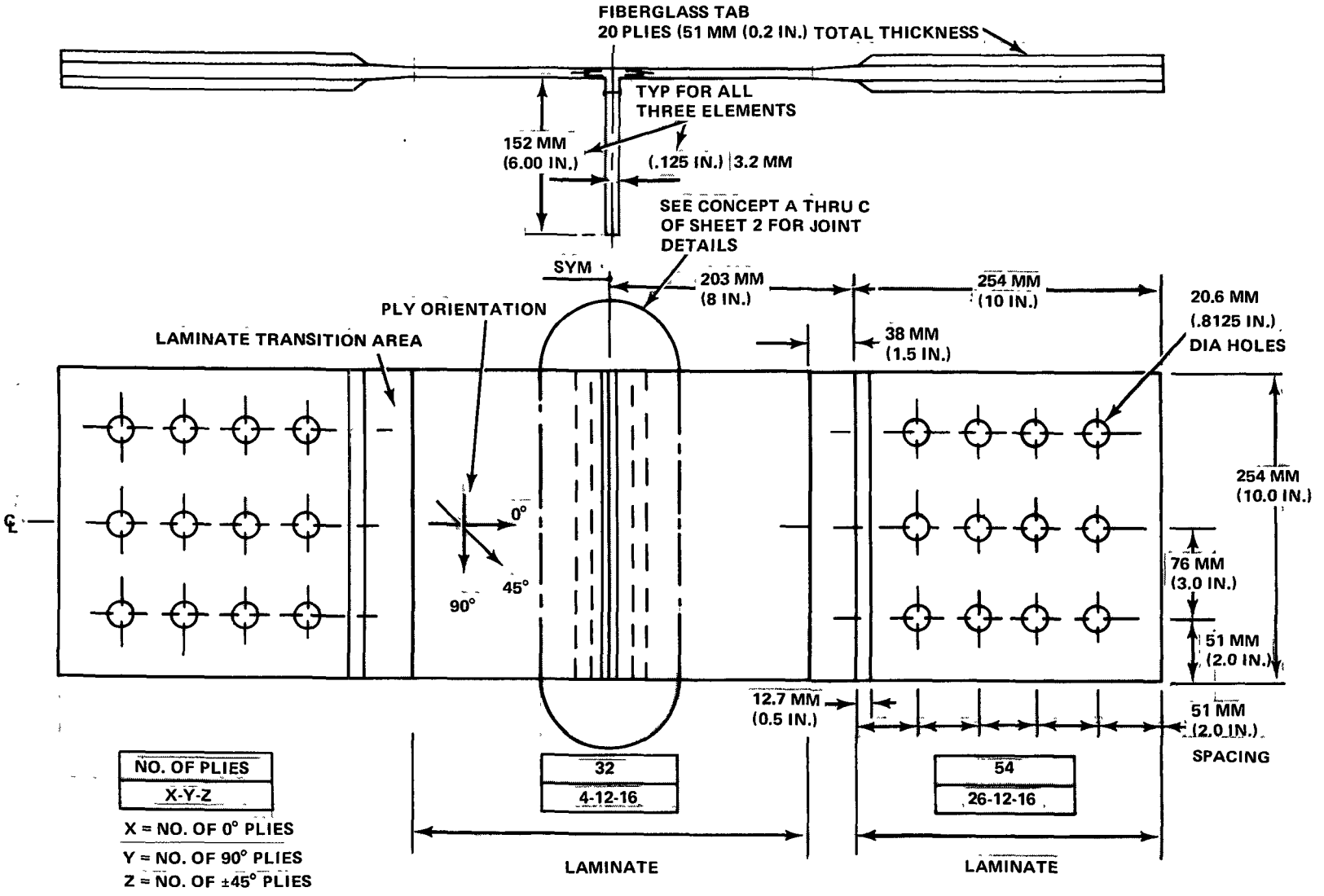

Figure 1 Specimen Configuration

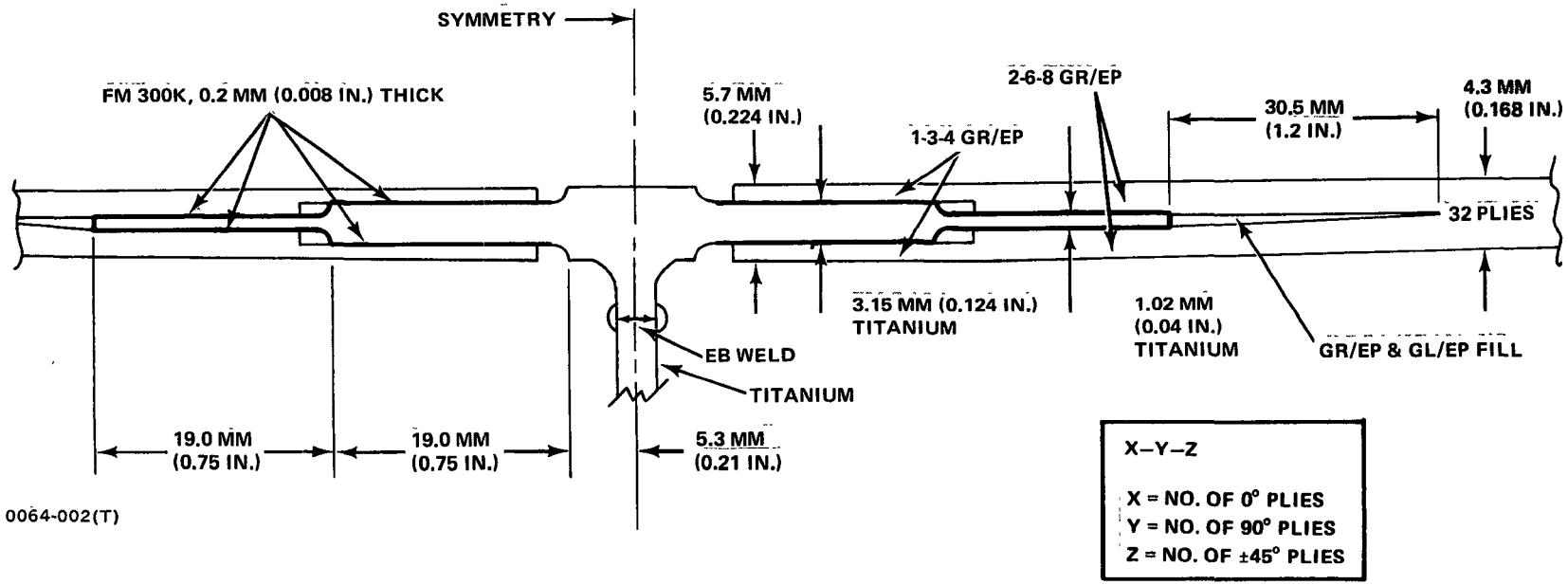

Figure 2 Concept A Detail, Two-Side Stepped Joint

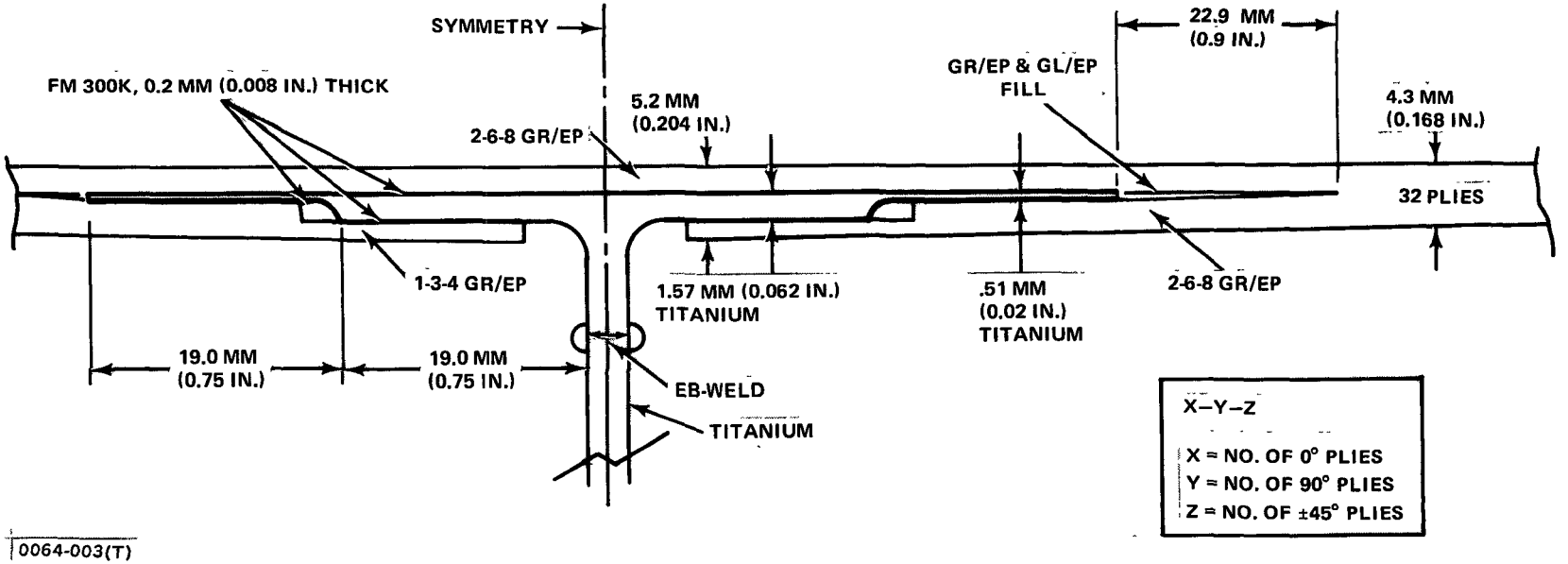

Figure 3 Concept B Detail, One-Side Stepped Joint

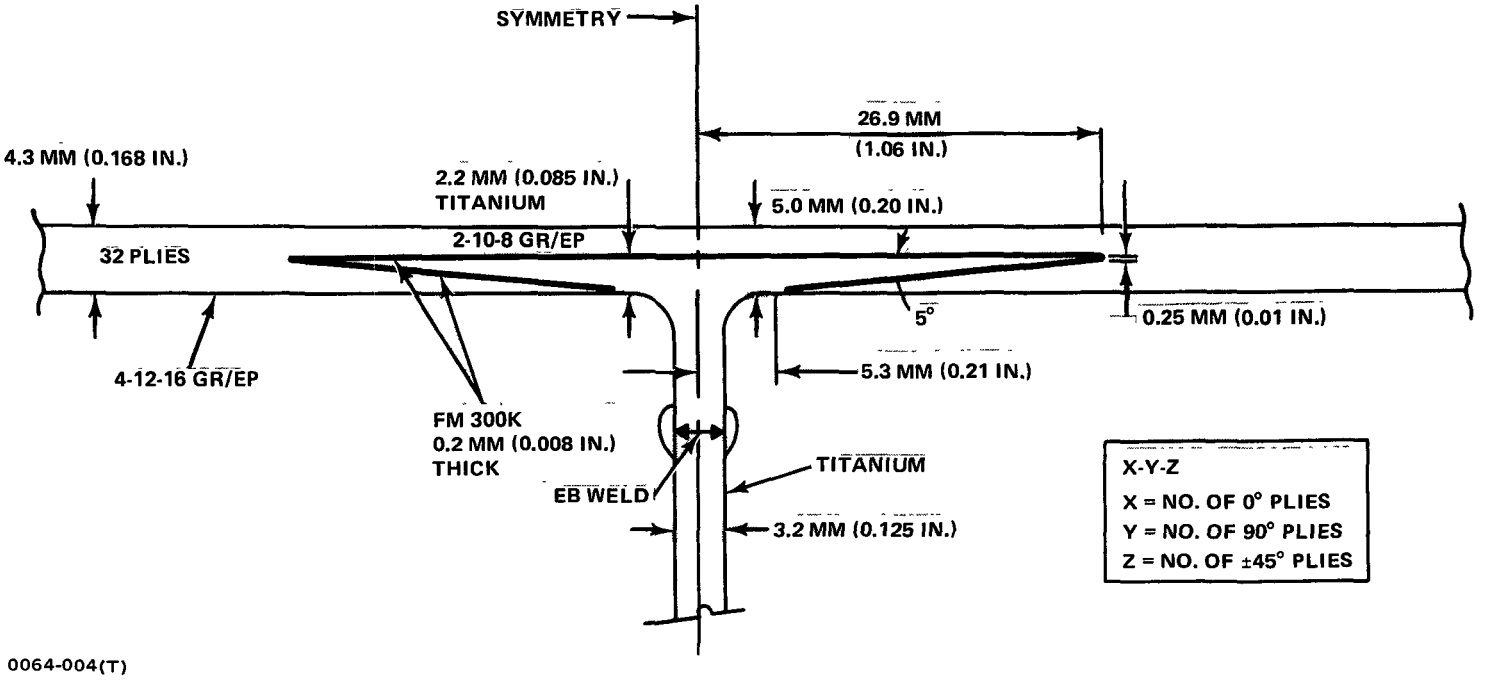

Figure 4 Concept C Detail, Scarf Joint

All three joint concepts were designed to fail in the adhesive since adhesive behavior under fatigue loading is the main program objective. In actual practice, bonded joints are usually designed to fail in one of the adherends since those types of failures are more predictable and thus can be better controlled.

Analysis

The three types of bonded joints shown in figures 2, 3, and 4 were analyzed for static load using Grumman's STEPS 7 computer program for Concepts A and B, and the method of reference 2 for Concept C.

STEPS 7, reference 3, is based on the closed-form modified elastic analysis of stepped bonded joints described in reference 4. The analysis has been periodically modified and includes thermal effects induced during cool-down from cure temperature. In addition, an arbitrary thickness of adherends can be input at each individual step; this implies that the joint need not be symmetrical. The only requirement is that the joint be adequately supported normal to the splice to minimize out-of-plane bending.

The adhesive normal and shear stress distributions are found by solving a set of simultaneous differential equations arising from the continuity of strains and displacements at the interfaces of the adhesives and adherends. The peak elastic stress is then reduced to account for the adhesive's nonlinear behavior. The relation between the elastic and plastic stress concentration factors is assumed to be

$$K_P = 1 + (K_E - 1)\, G_{SEC}/G_{TAN}$$

where K_E = Elastic stress concentration factor
K_P = Plastic stress concentration factor
G_{SEC} = Secant modulus at failure
G_{TAN} = Initial tangent modulus

A similar relation was used in reference 5 for notched plate test data showing good correlation. The validity of the above assumptions was verified on bonded joints with Metlbond 329 adhesive by Finite Element Analysis (FEA), nonlinear closed-form analysis, and test results.

The multiple-step joint takes the modified elastic solution of the single overlap splice and applies it separately to each step. The separate analyses at each step are combined by satisfying compatibility and equilibrium conditions at the internal discontinuities. Adherend bending is neglected in this analysis and, therefore, the adhesive normal stresses are assumed to be negligible. This last assumption was verified by FEA results which showed that, for a symmetric stepped lap joint, the maximum adhesive normal tension stress is less than 10% of the corresponding maximum shear stress. These normal stresses are low due to the symmetry of the joint configuration and the relatively small offset of load lines. This is also valid for scarf joints and, in fact, reference 2 neglects normal stresses.

Graphite/epoxy layer properties at room temperature used in the calculations were as follows:

f_1	(longitudinal failure stress)	= 1447 MPa (210 ksi)
E_{11}	(longitudinal modulus)	= 127.5×10^6 kPa (18.5×10^6 psi)
E_{22}	(transverse modulus)	= 11×10^6 kPa (1.6×10^6 psi)
τ_{12}	(shear failure stress)	= 68940 kPa (10000 psi)
G_{12}	(shear modulus)	= 5.86 kPa x 10^6 (0.85×10^6 psi)
μ_{12}	(Poisson's ratio)	= 0.25
α_{11}	(longitudinal coef of thermal expansion)	= 1.2×10^7 m/m/°C (2.2×10^{-7} in./in./°F)
α_{22}	(transverse coef of thermal expansion)	= 9.4×10^{-6} m/m/°C (1.52×10^{-5} in./in./°F)

Lamination theory with maximum stress failure criterion was used to obtain laminate properties from layer properties.

The adhesive properties for FM300K were obtained from American Cyanamid. The room temperature values used in the analysis were:

τ_{max} (shear peak failure stress) = 41360 kPa (6000 psi)
G_{TAN} (shear tangent modulus) = 882 MPa (128 ksi)
G_{SEC} (shear secant modulus) = 110 MPa (16 ksi)
γ_p
γ_e = 8, γ_p - plastic shear strain, γ_e, elastic shear strain
η (bondline thickness) = 0.20 mm (0.008 in.)

The static strength analysis of Concept A is straightforward using STEPS 7 because the joint is symmetric. The step length is reduced for analysis from 19 mm (0.75 in.) to 12.7 mm (0.5 in.) to account for manufacturing tolerances and titanium blade curvature. The output from the program (in English units) shown in figure 5 is for half the joint. The joint is predicted to fail in the adhesive at a load of 1026 kN/m (5860 lb/in.) at the edge of the first step or at the end of the zeroth step as identified in figure 5. The failure location is where the titanium insert is thinnest. This is within the desired limits. The predicted failure load in the titanium is 1443 kN/m (8246 lb/in.) and in the graphite/epoxy it is 1820 kN/m (10,400 lb/in.).

For static strength determination of the inserted one-sided stepped joint Concept B, figure 3, the STEPS 7 program was run twice. First, the upper bondline between the continuous part of the laminate and the titanium below the insert line was checked for strength. In this analysis the stiffness of the graphite/epoxy below the insert line was converted to an equivalent titanium stiffness. The results of the analysis are summarized in figure 6 and show the bondline strength to be more than adequate. Furthermore, 55.4% of the load is shed into the titanium insert. Thus, it was assumed in the strength analysis of the stepped bondline that 55.4% of the total specimen load goes through the lower composite-to-titanium joint. The output from this second analysis is summarized in figure 7. Again, the adhesive is the critical element with the total predicted fail load equal to 925 kN/m $\left(\frac{2929}{0.554} = 5287 \text{ lb/in.}\right)$.

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ANALYSIS OF STEPPED BONDED JOINT
COMPOSITE TO COMPOSITE OR COMPOSITE TO METAL

PROPERTIES AND CONFIGURATION


SPLICE IS IN TENSION

PROPERTIES OF MATERIAL NO. 1
----------------------------
LONGITUDINAL MOD.= 1260000   LONGITUDINAL STRENGTH= 110000
SHEAR STRENGTH= 10000   SHEAR MODULUS= 350000
TRANSVERSE MODULUS= 350000   POISSON'S RATIO=
THICKNESS= 5.25000E-03   ALPHA11= 2.20000E-06   ALPHA22= 1.25000E-05

PROPERTIES OF HOMOGENEOUS ADHEREND
----------------------------------
YOUNG'S MODULUS= 1600000   LONGIT. STRENGTH= 70000   SHEAR MOD.=
 620000
SHEAR STRENGTH= 7600   ALPHA= 5.50000E-06
STEP  1        THICKNESS= 0.02
STEP  2        THICKNESS= 0.02

PROPERTIES OF ADHESIVE
----------------------
GSEC GTAN= 0.12   BOND MOD.= 1200   BOND STR.= 6000
THICKNESS= 8.00000E-03

MATERIAL IN ZERO-DEG. DIRECTION-  NO. 1 COMPOSITE MATERIAL
MATERIAL IN 90-DEG DIRECTION---   NO. 1 COMPOSITE MATERIAL
MATERIAL IN + -45-DEG DIRECTION-  NO. 1 COMPOSITE MATERIAL

            COMPOSITE LAMINATE
            ------------------
STEP  ZERO-DEG.   90-DEG.   + -45 DEG.   STEP LENGTH        ALPHA11
      LAYERS      LAYERS      LAYERS        (IN)        (IN DEG F)
  1       2           6           8          0.5          2.85208E-06
  2       1           5           4          0.5          2.85208E-06

FACTOR OF SAFETY= 1    DELTA TEMP. FROM CURE=-200


OVERALL JOINT STRENGTH-AXIAL LOAD(LB IN)
                        ----------------
END OF
STEP  COMPOSITE  ADHESIVE  HOM. ADHER.  % LOAD IN N.A.  N(THERMAL)
  0       5217      3929                        0             0
  1       7018      6469       4123          0.600            33
  2                 7209       8060              1             0


OVERALL JOINT STRENGTH-SHEAR LOAD(LB IN)
                        ----------------
END OF
STEP  COMPOSITE  ADHESIVE  HOM. ADHER.  % LOAD IN N.A.  N(THERMAL)
  0       2191      3360                        0             0
  1       2408      5476       2788          0.545             0
  2                 4887       4712              1             0
```

0064-005(T)

Figure 5 STEPS 7 Output for Concept A

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SPLICE IS IN TENSION

PROPERTIES OF MATERIAL NO. 1
----------------------------
LONGITUDINAL MOD.= 18500000   LONGITUDINAL STRENGTH= 210000
SHEAR STRENGTH= 10000   SHEAR MODULUS= 850000
TRANSVERSE MODULUS= 1600000    POISSON'S RATIO= 0.25
THICKNESS= 5.25000E-03   ALPHA11= 2.20000E-07   ALPHA22= 1.52000E-05

PROPERTIES OF HOMOGENEOUS ADHEREND
----------------------------------
YOUNG'S MODULUS= 16000000   LONGIT. STRENGTH= 130000  SHEAR MOD.=
 6200000
SHEAR STRENGTH= 76000   ALPHA= 5.30000E-06
STEP  1         THICKNESS= 0.028
STEP  2         THICKNESS= 0.0484
STEP  3         THICKNESS= 0.0942
STEP  4         THICKNESS= 0.07

PROPERTIES OF ADHESIVE
----------------------
GSEC GTAN= 0.125   SHEAR MOD.= 128000  SHEAR STR.= 6000
THICKNESS= 8.00000E-03

MATERIAL IN ZERO-DEG. DIRECTION-  NO. 1 COMPOSITE MATERIAL
MATERIAL IN 90-DEG DIRECTION---   NO. 1 COMPOSITE MATERIAL
MATERIAL IN + -45-DEG DIRECTION-  NO. 1 COMPOSITE MATERIAL

            COMPOSITE LAMINATE
            ------------------
STEP  ZERO-DEG.   90-DEG.   + -45 DEG.   STEP LENGTH        ALPHA11
 0    LAYERS      LAYERS     LAYERS        (IN)          (IN DEG F)
 1        2           6          8          20            2.86296E-06
 2        2           6          8          0.539037717   2.86296E-06
 3        2           6          8          0.589339153   2.86296E-06
 4        2           6          8          20            2.86296E-06

FACTOR OF SAFETY= 1    DELTA TEMP. FROM CURE=-200


OVERALL JOINT STRENGTH-AXIAL LOAD(LB IN)
                        ----------------
END OF
STEP  COMPOSITE  ADHESIVE  HOM. ADHEP.  % LOAD IN H.A.  NX(THERMAL)
 0       5241      10508                    0              0
 1      12112      78134       6339         0.554          125
 2      17102      25394       9013         0.68           154
 3      19753    -319905       14896        0.723          164
 4                 21819       9100         1              0


OVERALL JOINT STRENGTH-SHEAR LOAD(LB IN)
                        ----------------
END OF
STEP  COMPOSITE  ADHESIVE  HOM. ADHEP.  % LOAD IN H.A.  NX(THERMAL)
 0       2182       8581                    0              0
 1       4224      53963       4401         0.483          0
 2       5705      21080       5957         0.617          0
 3       6507    -205120       9627         0.664          0
 4                 12404       5320         1              0

NX= 3000       NXX= 0         MARGIN OF SAFETY= 0.747
-------------------------
SPLICE IS CRITICAL AT END OF STEP 0     IN THE
COMPOSITE
```
0064-006(T)

Figure 6 STEPS 7 Output for the Upper Bondline of Concept B

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ANALYSIS OF STEPPED BONDED JOINT
COMPOSITE TO COMPOSITE OR COMPOSITE TO METAL

PROPERTIES AND CONFIGURATION


SPLICE IS IN TENSION

PROPERTIES OF MATERIAL NO. 1
----------------------------
LONGITUDINAL MOD.= 18000000   LONGITUDINAL STRENGTH= 210000
SHEAR STRENGTH= 10000   SHEAR MODULUS= 859000
TRANSVERSE MODULUS= 1600000    POISSON'S RATIO= 0.25
THICKNESS= 5.25000E-03   ALPHA11= 2.20000E-07   ALPHA22= 1.52000E-05

PROPERTIES OF HOMOGENEOUS ADHEREND
---------------------------------- ---
YOUNG'S MODULUS= 16000000   LONGIT. STREN-GTH= 130000  SHEAR MOD.=
 6200000
SHEAR STRENGTH= 76000   ALPHA= 5.30000E-06
STEP  1         THICKNESS= 0.02
STEP  2         THICKNESS= 0.062

PROPERTIES OF ADHESIVE
----------------------
GSEC GTAN= 0.125   SHEAR MOD.= 128000  SHEAR STR.= 11000
THICKNESS= 8.00000E-03

MATERIAL IN ZERO-DEG. DIRECTION-  NO. 1 COMPOSITE MATERIAL
MATERIAL IN 90-DEG DIRECTION---   NO. 1 COMPOSITE MATERIAL
MATERIAL IN + -45-DEG DIRECTION-  NO. 1 COMPOSITE MATERIAL

            COMPOSITE LAMINATE
            ------------------
STEP  ZERO-DEG.   90-DEG.   + -45 DEG.   STEP LENGTH        ALPHA11
      LAYERS      LAYERS      LAYERS        (IN)          (IN DEG F)
 1        2           6           8          0.5            2.85208E-06
 2        1           3           4          0.5            2.85208E-06

FACTOR OF SAFETY= 1    DELTA TEMP. FROM CURE=-200


OVERALL JOINT STRENGTH-AXIAL LOAD(LB IN)
                      ------------------
END OF
STEP  COMPOSITE  ADHESIVE  HOM. ADHEP.   % LOAD IN H.A.  NO. THERMAL
 0       5217       2929                      0                0
 1       7018       6469        4123          0.608           92
 2                  7209        8068          1                0


OVERALL JOINT STRENGTH-SHEAR LOAD(LB IN)
                       ----------------
END OF
STEP  COMPOSITE  ADHESIVE  HOM. ADHEP.   % LOAD IN H.A.  NO. THERMAL
 0       2191       3360                      0                0
 1       2408       5476        2788          0.545            0
 2                  4887        4712          1                0
```

0064-007(T)

Figure 7 STEPS 7 Output for the Stepped Bondline of Concept B

Parametric design curves from reference 2 were used to analyze Concept C, figure 4. However, STEPS 7 was run first as for the Concept B joint, to determine the percentage of load going through the lower composite/titanium interface. The output, figure 8, indicates that 47.2% of the total load is carried by the scarfed portion of the joint. The elastic-plastic joint strength was obtained using figure 6, p. 54 of reference 2. The effect of temperature is neglected since it is small for titanium/graphite/epoxy joints. The variables used to enter the parametric curves were as follows:

ℓ (effective bond length) = 16.5 mm (0.65 in.)

$$\lambda^2 = \frac{G_{TAN}}{\eta}\left[\frac{1}{E_1 t} + \frac{1}{E_2 t}\right] = 60.3$$

$\lambda\ell = 5.05$

where G_{TAN} = initial tangent modulus of adhesive

η = adhesive thickness

t = adherend thickness

E_1 = longitudinal modulus of graphite/epoxy laminate

E_2 = longitudinal modulus of titanium

For the nondimensional overlap, $\lambda\ell$, of 5.05 and a $\frac{\gamma_p}{\gamma_e}$ ratio of 8,

$$\frac{\tau_{avg}}{\tau_{max}} = 0.92$$

This translates into an axial failure load of 628 kN/m (3590 lb/in.) in the adhesive for the τ_{max} of 41260 kPa (6000 psi). The predicted failure load is lower than for that portion of the total laminate (2-ply 0°, 4-ply 90°, and 6-ply ±45°), which is calculated as 690 kN/m (3940 lb/in.). The total failure load on the joint (including the passing laminate on top of the insert) is then predicted to be 1330 kN/m $\left(\frac{3590}{0.472} = 7600 \text{ lb/in.}\right)$.

```
SPLICE IS IN TENSION

PROPERTIES OF MATERIAL NO. 1
----------------------------
LONGITUDINAL MOD.= 18500000   LONGITUDINAL STRENGTH= 210000
SHEAR STRENGTH= 10000   SHEAR MODULUS= 850000
TRANSVERSE MODULUS= 1600000    POISSON'S RATIO= 0.25
THICKNESS= 5.25000E-03   ALPHA11= 2.20000E-07   ALPHA22= 1.52000E-05

PROPERTIES OF HOMOGENEOUS ADHEREND
----------------------------------
YOUNG'S MODULUS= 16000000   LONGIT. STRENGTH= 130000  SHEAR MOD.=
 6200000
SHEAR STRENGTH= 76000   ALPHA= 5.30000E-06
STEP  1         THICKNESS= 0.0224
STEP  2         THICKNESS= 0.0365
STEP  3         THICKNESS= 0.0553
STEP  4         THICKNESS= 0.063

PROPERTIES OF ADHESIVE
----------------------
GSEC GTAN= 0.125   SHEAR MOD.= 129000  SHEAR STR.= 6000
THICKNESS= 8.00000E-03

MATERIAL IN ZERO-DEG. DIRECTION- NO. 1 COMPOSITE MATERIAL
MATERIAL IN 90-DEG DIRECTION---  NO. 1 COMPOSITE MATERIAL
MATERIAL IN + -45-DEG DIRECTION- NO. 1 COMPOSITE MATERIAL

            COMPOSITE LAMINATE
            ------------------
 STEP  ZERO-DEG.   90-DEG.    + -45 DEG.    STEP LENGTH        ALPHA11
       LAYERS      LAYERS      LAYERS          (IN)          (IN DEG F)
  1        2           10          8           20              3.73245E-06
  2        2           10          8           0.520344990     3.73245E-06
  3        2           10          8           0.567376857     3.73245E-06
  4        2           10          8           20              3.73245E-06

 FACTOR OF SAFETY= 1    DELTA TEMP. FROM CURE=-200


 OVERALL JOINT STRENGTH-AXIAL LOAD(LB IN)
                       ------------------
 END OF
 STEP  COMPOSITE  ADHESIVE  HOM. ADHER.   % LOAD IN H.A.  N.L. THERMAL
  0       5704      12069                      0                0
  1      10974      83248       6008           0.474           74
  2      14002      28903       7964           0.584           92
  3      16715     391464      10889           0.65           103
  4                 19228       8190           1                0


 OVERALL JOINT STRENGTH-SHEAR LOAD(LB IN)
                       -------------------
 END OF
 STEP  COMPOSITE  ADHESIVE  HOM. ADHER.   % LOAD IN H.A.  N.L. THERMAL
  0       2344       9767                      0                0
  1       3941      59586       4203           0.405            0
  2        365      25317       5365           0.517            0
  3       -775     260761       7161           0.566            0
  4                 11133       4788           1                0
0064-008(T)
```

Figure 8 STEPS 7 Output for the Upper Bondline of Concept C

The required thickness of the fiberglass/epoxy tabs was determined by treating the fiberglass as a nonlinear shear material between the 6.35-mm ($\frac{1}{4}$-in.) thick steel gripper plate and the composite. The analysis assumes that the pressure plates provide adequate compressive forces to eliminate peel stresses. Using this analysis, a 5.1 mm (0.2 in.) thickness of fiberglass/epoxy was required, based on the interlaminar shear capability as the failure criterion.

SPECIMEN FABRICATION

Manufacturing Flow

The manufacturing flow for the test specimens, figure 9, was based on recent Grumman experience in producing specimens of similar configuration. The titanium fittings were fabricated as welded detail parts prior to the composite laminating step in preference to welding after cocure/bonding. This approach eliminated the possibility of thermal degradation/stressing of the composite and bond due to the heat generated during welding. For the same reason, the titanium blades for each specimen type were cut to final length (figures 10, 11, and 12) and cocure-bonded into the composite laminate as details separated by graphite/epoxy fillers in preference to requiring cutting of the titanium while trimming individual specimens from a continuous laminate. In this approach, the individual specimens were readily parted using a water-cooled diamond grit blade. In contrast, the titanium/graphite/epoxy stackup can be sawed only by a friction blade which generates sufficient heat to discolor the titanium and possibly degrade both the laminate and adhesive locally.

The composite laminate was laid-up on an aluminum tool using mylar templates. The tool was recessed in the gripper areas to accommodate the additional plies needed in those areas and hence keep the load line concentric. After all the graphite/epoxy plies, glass/epoxy wedges, and adhesive-coated titanium details were laid-up on the tool, the bleeder system was applied and the assembly cured. Standard 177°C cure cycle was used. A completed assembly from which six specimens were cut is shown in figure 13.

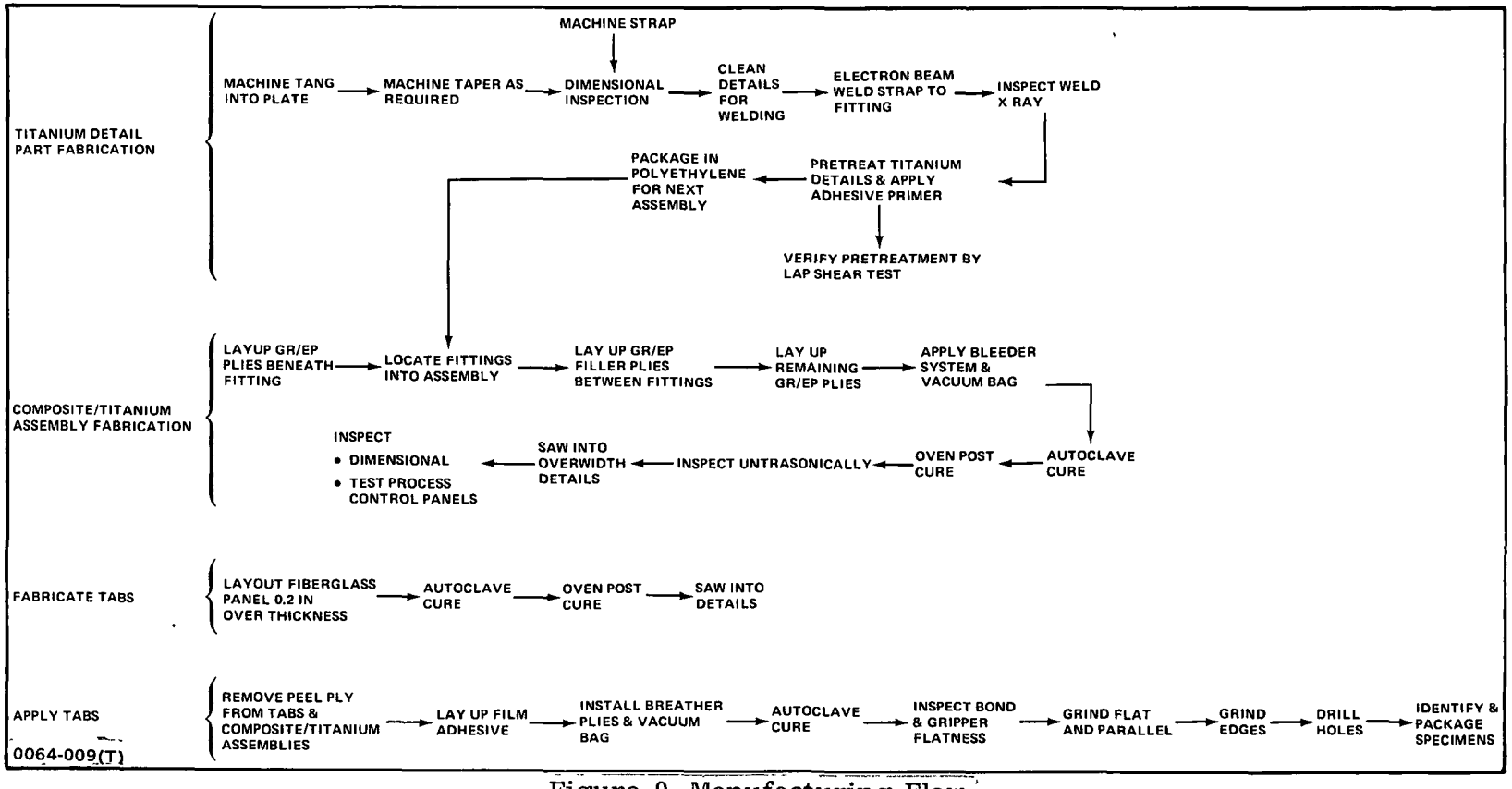

Figure 9 Manufacturing Flow

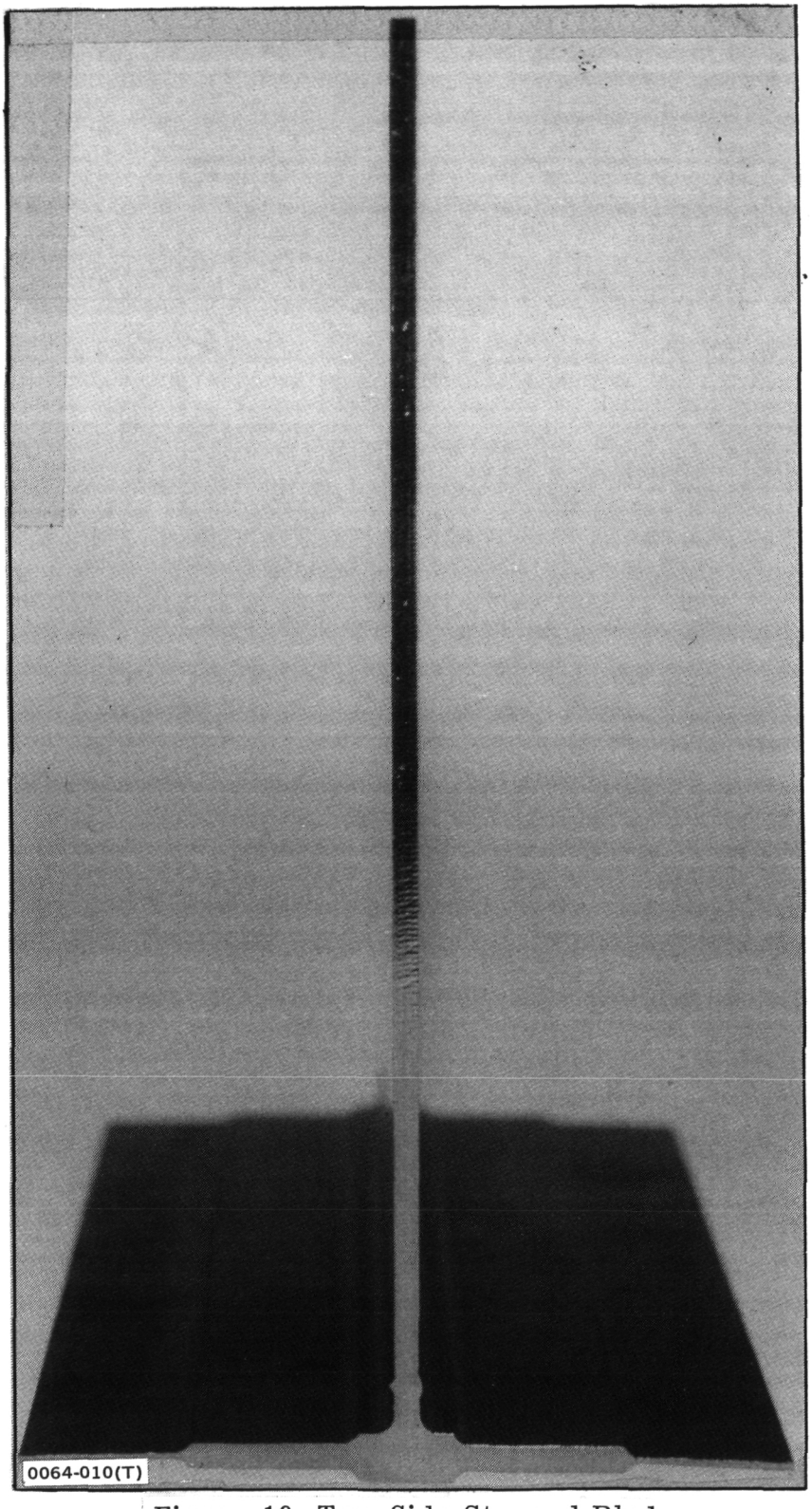

Figure 10 Two-Side Stepped Blade

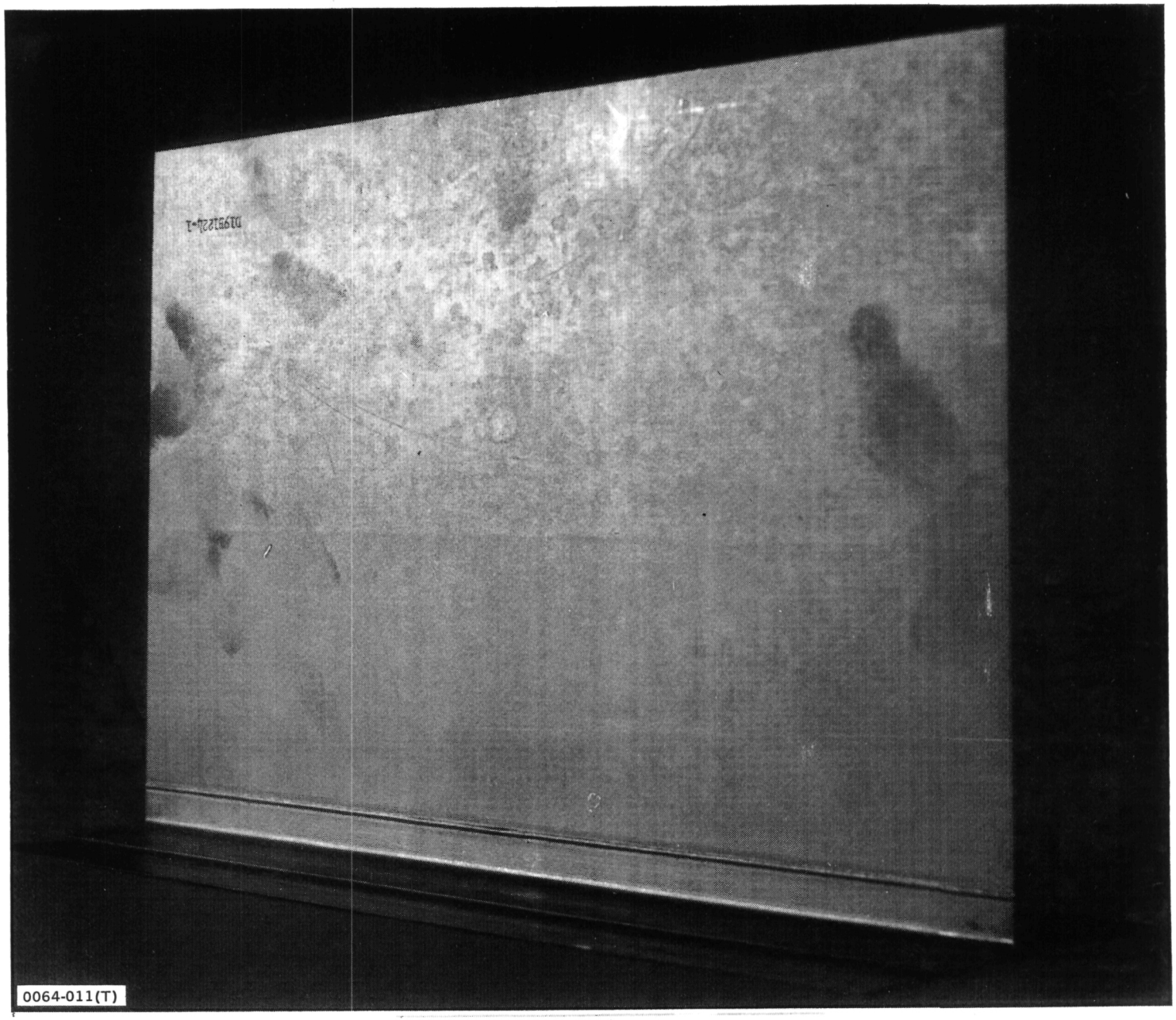

Figure 11 One-Side Stepped Blade

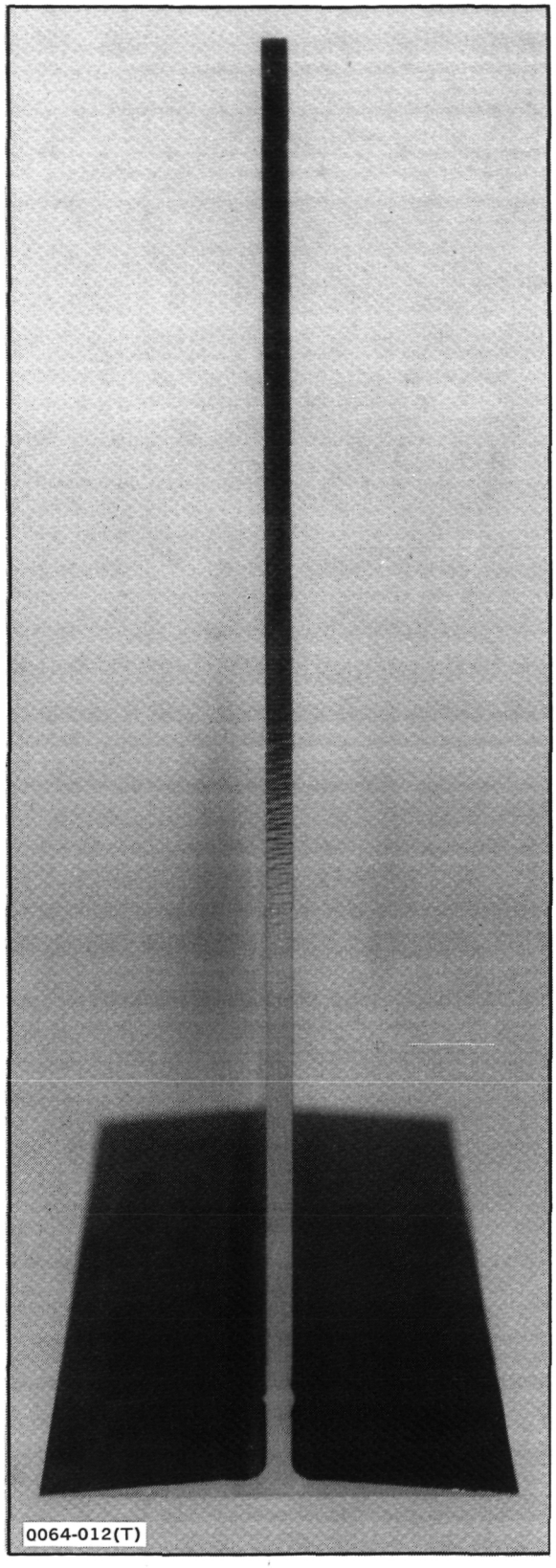

Figure 12 Scarfed Blade

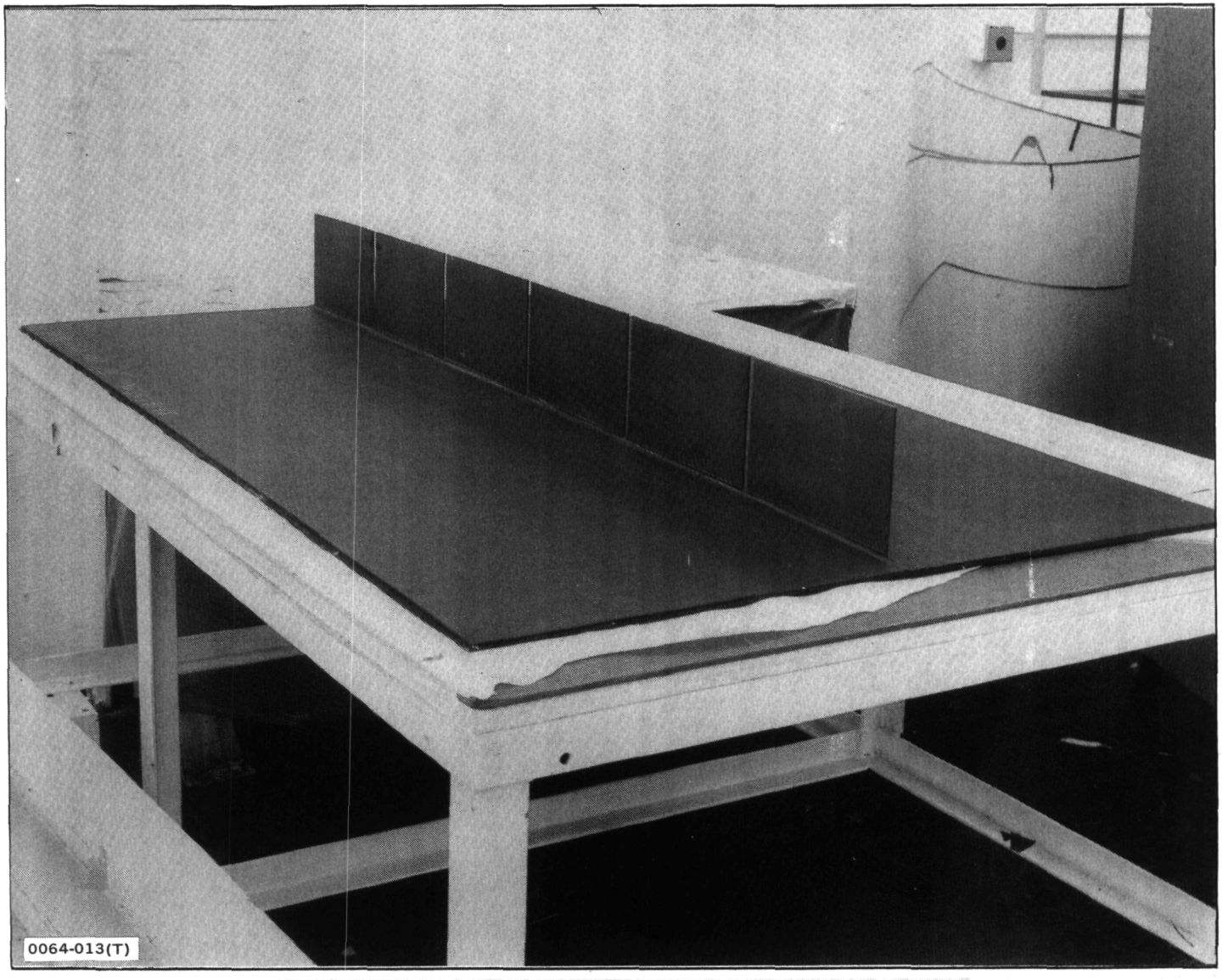

Figure 13 Completed Typical Bonded Joint Assembly

The grippers were machined flat and parallel after bonding to provide a load path through the center of the specimen. The specimen edges were ground to provide a clean and visible titanium-to-composite interface. Lastly, the required holes were drilled with the aid of a template.

Pretreatment Requirements

The titanium pretreatment for this program was identical to that used in the boron/epoxy-to-titanium splice joints in the F-14 horizontal stabilizer skins and consisted of the following steps:

(1) The surface was cleaned and solvent-wiped using MEK or isopropyl alcohol.

(2) All surfaces were pressure-blasted using virgin aluminum oxide grit with a regulated and controlled air pressure.

(3) Parts were spray or pressure-rinsed with tap or deionized water at temperatures below 38°C and checked for water break-free surfaces.

(4) The parts were immersed for 10 to 15 min in a Pasa Jell 107M control etchant-conversion coating bath within 4 hours of grit blasting.

(5) The parts were water-rinsed in a two-step operation.

(6) Parts were force-air-dried using clean filtered air at temperatures below 38°C.

(7) Within 2 hr, EC-2333 bonding primer was applied and oven-dried.

The pretreatment sequence was performed as a continuous operation 2-3 days before the titanium details were placed in the assembly and the adhesive applied.

QUALITY CONTROL

Production quality control methods were used to ensure that all composite/metallic specimens were of a consistent level of quality. Quality was controlled by incorporating predetermined inspection points on the work orders, facilitating early detection of defects and minimizing production variables. Quality of fabricated specimens was determined by NDE methods which provided maximum defect resolution. Specific quality control/NDE steps employed during specimen manufacture are described below.

Vendor Certification of Materials

Material suppliers were required to certify all incoming material met the requirements of the appropriate Grumman specification: GM3013 for graphite/epoxy and GM3103 for 6Al-4V titanium alloy.

Receiving Inspection

Quality Control verified the quantity, condition, and conformance of incoming material to specification and procurement requirements. Selective samples were taken and sent to the laboratory for physical and mechanical determinations.

Process Control

The graphite/epoxy layup was controlled through the use of mylar templates having all the required information for each particular ply. Each ply was then inspected for gaps, overlaps, orientation, and inclusions. Cure cycles were monitored for conformance by permanent recordings of vacuum, pressure, temperature, and time.

Process Verification

The bonding process was verified by:

- Five titanium lap shear specimens (figure 14) processed with each group of details during the titanium pretreatment
- Five titanium/concured bonded to graphite/epoxy lap shear specimens, figure 14, processed with each assembly during the cure.

No shear strength requirements for these specimens were established since Grumman has no previous production history with the FM300K adhesive. The results of these tests are given in table 1. The test specimens strength was judged sufficiently high and reasonably in agreement with the pure shear strength value of 41,360 mPa (6000 psi) used in the analysis. The lower

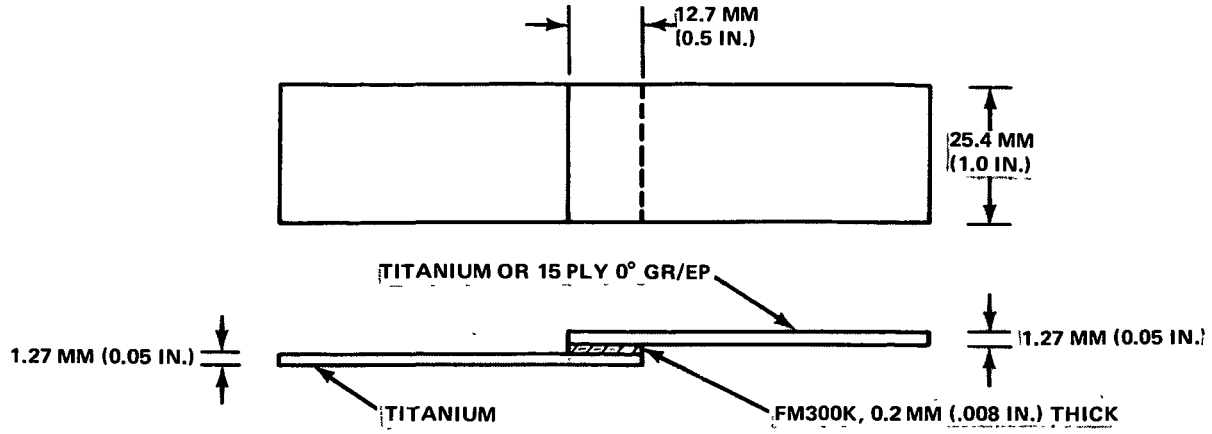

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Figure 14 Verification Lap Shear Specimen

Table 1 Results of Process Verification Lap Shear Tests

LAMINATE ASSEMBLY DRAWING NO.	JOINT STRENGTH — kPa, (psi)				
	TITANIUM TO TITANIUM			TITANIUM TO GR/EP	
	RT	121°C (250°F)	177°C (350°F)	RT	121°C (250°F)
D19B1220	34600	26750	–	32810	21790
	(5020)	(3880)	–	(4760)	(3160)
	36100	24820	–	31850	26610
	(5240)	(3600)	–	(4620)	(3860)
	–	23850	–	–	22340
	–	(3460)	–	–	(3240)
D19B1221	40950	–	9240	39020	26470
	(5940)	–	(1340)	(5660)	(3840)
	37230	–	10750	37500	24960
	(5400)	–	(1560)	(5440)	(3620)
	–	–	6000	–	24130
	–	–	(870)	–	(3500)
D19B1222	33230	25650	–	18900*	16000
	(4820)	(3720)	–	(2740)*	(2320)
	38330	21510	–	22890*	18610
	(5560)	(3120)	–	(3320)*	(2700)
	–	22340	–	–	16820
	–	(3240)	–	–	(2440)
AVERAGE	36750	24150	8670	35300	22000
	(5330)	(3503)	(1257)	(5120)	(3187)

*LOW VALUES, NOT INCLUDED IN AVG. SUSPECT THE FAILURE OCCURED IN COMPOSITE AND NOT IN ADHESIVE.

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strengths obtained here with a 12.7 mm (0.5 in) overlap specimen are caused by additional peel stresses existing in this type of specimen.

In addition to bonding process verification tests, process control test panels were used to verify the adequacy of the cure cycles. These panels were fabricated with each autoclave cure cycle and submitted to the Quality Control Laboratory with an accompanying traveler which identified the represented specimens, including batch and roll number(s) of the material. The control panel was machined into test coupons which were subjected to both flexural strength and modulus tests and horizontal shear strength tests at room temperature and 177°C. The results of these tests are included as table 2.

Table 2 Test Results of Cure Verification Specimens

LAMINATE ASSEMBLY DRAWING NO.	STRENGTH MPa, (ksi)			
	FLEXURE(1)		HORIZONTAL SHEAR(2)	
	RT	177°C (350°F)	RT	177°C (350°F)
D19B1220	2082 (302) 2193 (318) 2020 (293)	1365 (198) 1289 (187) 1551 (225)	110 (16.0) 112 (16.2 114 (16.6)	70.3 (10.2) 68.3 (9.9) 68.9 (10.0)
D19B1221	1779 (258) 2076 (301) 1772 (257)	1455 (211) 1462 (212) 1441 (209)	113 (16,4) 114 (16.5) 126 (18.3)	60.0 (8.7) 63.4 (9.2) 59.3 (8.6)
D19B1222	1806 (262) 1738 (252) 1634 (237)	931* (135) 958* (139) 979* (142)	68.9 (10.0) 59.3 (8.6) 59.3 (8.6)	35.2 (5.1) 36.5 (5.3) 44.8 (6.5)

(1) FLEXURE SPECIMENS WERE UNIDIRECTIONAL (0°) LAMINATE – 12.7 x 114.3 mm (0.5 x 4.5 in.) 15-PLY THICK

(2) HORIZONTAL SHEAR SPECIMENS WERE UNIDIRECTIONAL (0°) LAMINATE – 6.35 x 15.2 mm (0.25 x 0.6 in.), 15-PLY THICK

* BELOW SPECIFICATION, THE QC PANEL WAS OVERTHICK AND POROUS. HOWEVER, THE LAMINATE ASSEMBLY WAS WITHIN THICKNESS TOLERANCE. SUSPECT VACUUM LEAKAGE IN QC PANEL.

Nondestructive Inspection

Each laminate/joint assembly was ultrasonically inspected before cutting the laminate into individual specimens. Ultrasonic inspection was selected since it is the primary technique for detecting voids, delaminations, and inclusions and is part of standard inspections used at Grumman. The inspection was first conducted using the Immersion-Through-Transmission (Reflector Plate) Technique. This immersion technique utilizes automated scanning and provides a rapid means of 100% inspection, but does not provide depth of defect information. For defect depth information (in cases where defects were found), the contact transducer resonance technique was used. These techniques are used on the F-14 Horizontal Stabilizer covers with excellent results.

Representative standards, designed and fabricated by Grumman, were utilized for ultrasonic detection of internal defects. These standards contained designed defects of the critical flaw size or smaller. All discrepancies detected by NDI were reported to cognizant development engineer on a standard Grumman discrepant material report (DMR) form.

Internal defects were found only on one laminate assembly. These were reported on the DMR form and consisted of some minor porosity in the grip area and a disbond in the adhesive joint. The porosity was deemed not critical for a tension specimen. The size and location of the disbond affected only one specimen out of six and is shown in figure 15. The disbond, located between the composite and titanium on the upper bond side, was not repairable since any injections of adhesive would open-up the separation. The length of the disbond was verified visually by polishing the edge.

Each of the panels with six titanium blades were slightly bowed due to thermal stresses induced during molding. The panels could be flattened with minimal applied load and are satisfactory for tensile testing. This type of bowing is expected on flat panels and generally is not noted in formed production wing covers.

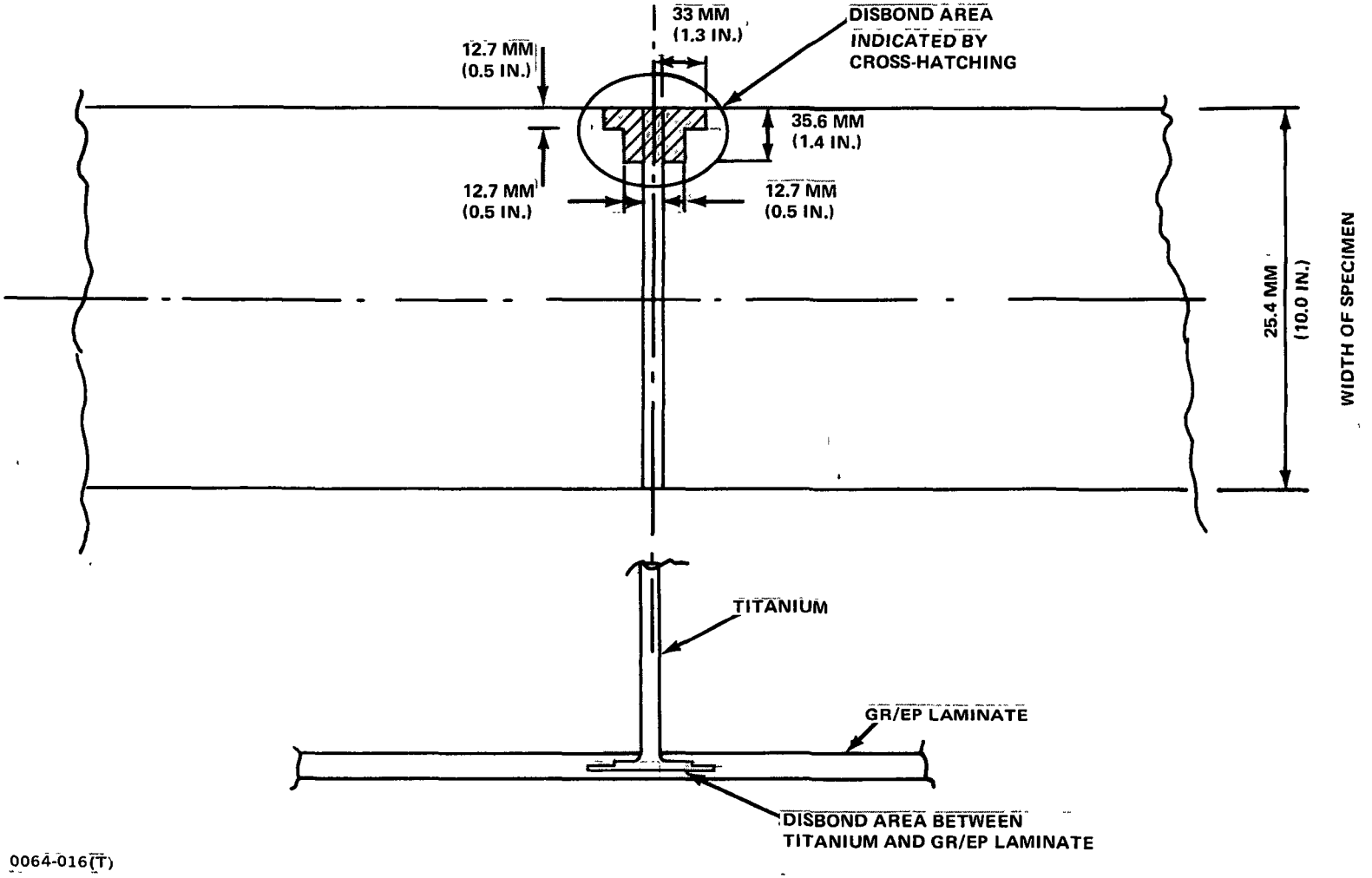

Figure 15 Ultrasonically Detected Void Area

CONCLUSIONS

Eighteen bonded-joint test specimens representing three different designs of a composite chordwise-bonded splice were designed and fabricated using current aircraft industry practices.

The three types of joints are:

- Full wing laminate penetration, two-side stepped;
- Mid-thickness penetration, one side stepped;
- Partial penetration, scarfed.

These were analyzed using state-of-art elastic joint analysis modified for plastic behavior of the adhesive. The static tensile fail load at room temperature was predicted to be between 925 and 1330 kN/m (5287 and 7600 lb/in.) with critical location in the adhesive.

There are certain shortcomings in the analysis that make the failure predictions less reliable than on might have expected. First, the properties of the adhesive are not characterized to an extent that statistically defined strength and stiffness values are available. Second, for the inserted titanium blade joints, a more refined finite element analysis would be required to determine the exact distribution of load between the passing composite laminate and the composite/metal joint. Lastly, for the scarfed joint, the analysis is based on a knife-edge end of the titanium blade which, in practice, is never realized.

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1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR-166099		

4. Title and Subtitle	5. Report Date
DESIGN AND FABRICATION OF REALISTIC ADHESIVELY BONDED JOINTS	May 1983
	6. Performing Organization Code
7. Author(s)	8. Performing Organization Report No.
Peter Shyprykevich	
	10. Work Unit No.
9. Performing Organization Name and Address	
Grumman Aerospace Corporation Bethpage, NY 11714	11. Contract or Grant No. NAS1-16952
	13. Type of Report and Period Covered Contractor Report
12. Sponsoring Agency Name and Address	
National Aeronautics and Space Administration Washington, DC 20546	14. Sponsoring Agency Code

15. Supplementary Notes Langley technical monitor: R. A. Everett, Jr.
The contract research effort which led to the results in this report was financially supported by the Structures Laboratory, U.S. Army Research and Technology Laboratories (AVRADCOM), NASA Langley Research Center.

16. Abstract

Eighteen bonded joint test specimens representing three different designs of a composite wing chordwise bonded splice were designed and fabricated using current aircraft industry practices.

Three types of joints (full wing laminate penetration, two-side stepped; mid-thickness penetration, one-side stepped; and partial penetration, scarfed) were analyzed using state-of-the-art elastic joint analysis modified for plastic behavior of the adhesive. The static tensile fail load at room temperature was predicted to be:

- 1026 kN/m (5860 lb/in) for the two-side stepped joint
- 925 kN/m (5287 lb/in) for the one-side stepped joint
- 1330 kN/m (7600 lb/in) for the scarfed joint

All joints were designed to fail in the adhesive.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Adhesively bonded joints Structural design Graphite/epoxy Computer analysis Wing-type chordwise splice	Unclassified - Unlimited Subject Category 39

19 Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
Unclassified	Unclassified	31	

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